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Mark Sumich, Ames Research Center, Moffett Field, California
Keith T. Kedward, University of California, Santa Barbara, California

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**Ames Research Center**
Moffett Field, California 94035-1000

# DEVELOPMENT OF A FATIGUE-LIFE METHODOLOGY FOR COMPOSITE STRUCTURES SUBJECTED TO OUT-OF-PLANE LOAD COMPONENTS

Mark Sumich and Keith T. Kedward*

Ames Research Center

## SUMMARY

This report documents the efforts of the RSRA/X-Wing Project Office to identify and implement a fatigue-life methodology applicable to demonstrated delamination failures for use in certifying composite rotor blades. The RSRA/X-Wing vehicle was a proof-of-concept stopped rotor aircraft configuration which used rotor blades primarily constructed of laminated carbon fiber. Delamination of the main spar during ground testing demonstrated that significant interlaminar stresses were being produced. Analysis confirmed the presence of out-of-plane load components. A review of the available failure methodologies was undertaken to determine the approach most applicable to certifying primary composite structures which can fail via delamination. The final selection of the "wear out" (residual strength) methodology and the requirements for its implementation are discussed. Composite structural designs incorporating out-of-plane load components are not endorsed. However, a means of certifying development hardware is necessary when an identification of matrix dominated failure modes occurs late in the development cycle. Efforts are continuing at NASA Ames Research Center to assess the validity of the "wear out" based failure methodology for interlaminar tension failures. A different type of test coupon has been developed and testing is underway to establish a database on the "wear out" characteristics of a composite structure subjected to out-of-plane loads.

## INTRODUCTION

The advantages of fiber reinforced materials are well known to many designers of aerospace structures. The inherent anisotropy of composites results in exceptional performance along fiber directions, providing unsurpassed specific strength and stiffness. These benefits can enable the design of structures which would not be practical with the exclusive use of isotropic metallics, e.g., forward swept wings. The successful designer, however, must have a clear understanding of the characteristics peculiar to fiber reinforced materials.

The failure modes observed in composites are substantially different from those seen in metallics. Delamination is one failure mechanism unique to laminated composites which is often characterized by high rates of propagation and catastrophic consequences. Numerous occurrences of delamination failures during the development of composite hardware have been documented (refs. 1

*University of California, Santa Barbara, CA.

and 2). Delamination is caused by excessive stress within the matrix of a composite material. One of these matrix stresses is referred to as interlaminar tensile stress (ILTS), short transverse stress, or through-the-thickness stress. Primary fiber-carried loads in regions of curvature, ply dropoffs (ref. 3), or ply waviness can create secondary stresses within the matrix which tend to pull the plys of the laminate apart. Most practical composite structures contain a number of these and other features which can encourage delamination (fig. 1). Since the strength of the matrix may be two orders of magnitude less than the strength of the fiber, even low levels of interlaminar tension (6.89 Mpa, 1.00 ksi) may jeopardize the ability of the laminate to sustain its design load. Concern regarding the presence of out-of-plane load components has prompted the U.S. Navy and FAA to award an R & D contract to McDonnell Aircraft and Northrop (ref. 4).

The authors wish to acknowledge the significant contributions of R. S. Wilson and S. M. Ehlers in evaluating the application of the damage tolerance and safe life/reliability failure methodologies to matrix-dominated failure modes.

## SYMBOLS

| | |
|---|---|
| $\alpha$ | Weibull shape parameter |
| $\beta$ | Weibull scale parameter |
| $\alpha_S$ | static strength shape parameter |
| $\alpha_L$ | fatigue life shape parameter |
| $\alpha_R$ | residual strength shape parameter |
| $\sigma_{zz_{max}}$ | peak interlaminar tensile stress |
| F(0) | initial residual (static) strength |
| F(t) | residual strength after time t |
| F(xx) | minimum required residual strength |
| F(PL) | proof load level |
| $F_{max}$ | maximum fatigue spectrum stress |
| $G_{IC}$ | mode I strain energy release rate |
| $G_{IIC}$ | mode II strain energy release rate |
| a | crack length |

| | |
|---|---|
| N | no. of fatigue cycles |
| t(op) | desired safe operating period |
| r | wear out parameter |
| $A_4$ | material and history dependent constant |
| R | damage accumulation rate, $(r - 1)\ A_4\ F_{max}^{2r}$ |
| P | applied load |
| L | moment arm |
| c | thickness at test section |
| b | width at test section |
| $R_i$ | inside radius of laminate at test section |
| $R_o$ | outside radius of laminate at test section |

## COMPOSITE ROTOR BLADES

The RSRA/X-Wing vehicle was a proof-of-concept stopped rotor aircraft configuration which utilized a modified Rotor Systems Research Aircraft (RSRA) to examine the performance of a circulation controlled four-bladed X-Wing rigid rotor (fig. 2). The X-Wing rotor may be operated in a rotary wing mode much like a conventional helicopter or flown in a stopped rotor mode as fixed, X-shaped wings. The lower, conventional wing is part of the basic RSRA and it permits the aircraft to share the lift between the X-Wing rotor and the main wing or explosively jettison the X-Wing rotor entirely and return to land safely. A complete description of the RSRA/X-Wing aircraft and a review of the overall program are presented in reference 5. The program was terminated after the initiation of rotorless flight testing due to insufficient funds.

A schematic of the X-Wing rotor system and a breakdown of the major precured blade components are shown in figures 3(a) and 3(b). Each of the four 600-lb. rotor blades was constructed almost entirely out of carbon fiber. The primary spar in each blade is a 260-lb laminated carbon fiber I-beam called the flexbeam. The flexbeam attaches to the rotor hub at its root end and to the outboard I-beam at its tip and carries the flatwise and edgewise bending moments and vertical shear loads into the hub. The flexbeam has build-ups in flange thickness up to 2 in. thick at the root end to reinforce the flexbeam where it is bolted to the hub, and tip end flange build-ups form wide, flat surfaces suitable for bonding into the outboard portion of the blade. Since the flexbeam is not bonded along its length to the C-sections and is designed to be torsionally compliant, it is free to twist as the rest of the bonded blade assembly is rotated around it to provide collective pitch control.

The material used was Celion G40-600 fiber impregnated with Narmco 5245C epoxy modified bismaleimide resin to withstand the high-temperature compressed air that was internally channelled down the leading and trailing edge ducts of each blade to provide aerodynamic circulation control. All components were cured at 350° F, then post-cured at 400° F to provide the necessary high temperature capability. The ducts would regularly be exposed to 320° F air which results in a 225° F operating environment at the location of the flexbeam.

## Spar Delamination

Two isolated flexbeams were selected for testing to assess the overall design approach and the effects of manufacturing variations on structural integrity. The flexbeams were bolted to a simulated hub and bending and torsion loads were applied at the tip end by hydraulic cylinders. The flexbeams were extensively strain gauged and also incorporated an experimental fiber-optic crack detection system.

Flexbeam serial number 3 was tested statically to failure at room temperature. The load was gradually increased until a delamination failure suddenly announced itself with a loud bang. The delamination occurred in the upper root end flange which was exposed to high levels of axial tension (fig. 4). Following the failure, the flexbeam continued to carry the load in the same direction with very little change in stiffness. When the loading was reversed, the delaminated flange buckled catastrophically before achieving the previous load level. After reviewing the strain gauge data it was determined that the flexbeam had demonstrated the required static margin of safety above anticipated stopped rotor flight loads, but had failed before demonstrating a sufficient static margin above the flight loads expected for the conversion process from stopped rotor to rotary wing flight and back.

Flexbeam serial number 6 was then installed in the test fixture and subjected to spectrum and constant amplitude oscillatory loads at its anticipated operating temperature of 225° F. The spectrum loading corresponded to conversion flight loads and was demonstrated for the equivalent of 120 conversions between stopped rotor and rotary wing flight. The flexbeam was then subjected to blocks of constant amplitude oscillatory load applied in increasing order of magnitude and failed at 2880 cycles corresponding to peak conversion flight load. The type of failure was the same as in the previous static test of flexbeam number 3 and confirmed the existence of a matrix dominated failure mode.

These events prompted further analysis of the levels of interlaminar stresses present in the flexbeam and other blade components. A 3-D NASTRAN analysis verified the presence of high levels of interlaminar stress (tension and shear) in the flexbeam's flange buildup areas and indicated that some of the bondlines between components would also be exposed to significant interlaminar stresses.

## CANDIDATE CERTIFICATION APPROACHES

A Government/contractor team was assembled to assess the current state-of-the-art in failure methodologies and recommend a procedure to certify the rotor blades for flight. In the literature, there are currently no validated methodologies for the certification of primary composite structure which can fail via interlaminar tension and/or shear. To date no certification requirements analogous to the damage tolerance approach used for metallic structures (Mil-A-83444) have been approved for composites. However, a variety of approaches have been proposed and are currently under consideration. Three candidate certification approaches applicable to composite airframe structures are reviewed in the following sections with regard to their suitability for the certification of the X-Wing composite rotor blades. The three approaches considered were: (1) damage tolerance, (2) safe life (reliability), and (3) wear out model.

### Damage Tolerance Methodology

The damage tolerance methodology assumes that the largest undetectable flaw exists at the most critical location in the structure and that structural integrity is maintained through flat growth until detected by periodic inspection (ref. 6). The approach is based on the draft of the proposed Air Force Damage Tolerance specification requirements for organic matrix aircraft structure prepared by Northrop/Boeing as joint contractors under the Air Force Damage Tolerance of Composites contract F33615-82-C-3213.

In this approach, damage tolerance capability covering both flaw growth potential and residual strength is verified by both analysis and test. The analysis would assume the presence of a flaw or damage placed at the most unfavorable location and orientation with respect to applied loads and material properties. The assessment of each component should include areas of high strain, strain concentrations, minimum margin of safety details, major load path, damage prone areas, and special inspection areas. The structure selected as critical by this review should be considered for inclusion in the experimental and test validation of the damage tolerance substantiation procedures. Those structural areas identified as critical after the analytical and experimental screening should form the basis for the subcomponent and full scale component validation test program. Test data on the coupon, element, detail subcomponent, and full scale component level, whichever is applicable, should be developed or be available to: (1) verify the capability of the analysis procedure to predict damage growth or no growth, and residual strength; (2) determine the effects of environmental factors; and (3) determine the effects of repeated loads.

Flaws or damage will be assumed to exist initially in the structure as a result of the manufacturing process or to occur at the most adverse time after entry into service. The specific flaw or damage size requirements are as follows.

**Scratches** Assume the presence of a surface scratch that is 4.0 in. in length and 0.02 in. deep.

**Delamination** Assume the presence of an interply delamination that has an area equivalent to a 2.0 in. diameter circle with dimensions most critical to its location.

**Impact Damage** Assume the presence of damage caused by the impact of a 1.0-in. diameter hemispherical impactor with 100 ft-lb of kinetic energy or with that kinetic energy required to cause a dent of 0.10 in. deep, whichever is least.

Where initial flaw or damage assumptions for safety of flight structure are less than above, a nondestructive inspection (NDI) demonstration shall be performed. This demonstration shall verify that all flaw or damage greater than the assumed flaw or damage size will be detected with a statistical confidence of 95% and a statistical probability of 90%.

Component, assembly, or complete airframe inspection proof tests of every airframe shall be performed whenever special nondestructive inspections cannot be validated and initial flaw or damage assumptions for damage tolerant structure are less than specification requirements. The purpose of this testing shall be to define maximum possible initial flaw size or other damage in that portion of the structure without multiple load paths or provisions for flaw or damage growth arrest.

A decision to employ proof testing must take the following factors into consideration.

1. The loading that is applied must accurately simulate the peak stresses and stress distributions in the area being evaluated.

2. The effect of the proof loading on other areas of the structure must be thoroughly evaluated.

3. Local effects must be taken into account in determining the maximum possible initial flaw or damage size after test and in determining subsequent flaw or damage growth.

An analytical technique for the evaluation of growth or no growth of delaminations is an essential tool for an evaluation of the damage tolerance of composite structures. A numerical method is available which uses finite element analysis and a crack closure integral technique from fracture mechanics (ref. 7). Prerequisites for an evaluation are (1) a structural analysis made in sufficient detail to indicate locations where critical interlaminar stresses exist, (2) test-derived critical interlaminar strain energy release rates $G_{IC}$, $G_{IIC}$ and a subcritical growth law, i.e., da/dN versus $\Delta G$ for each mode, and (3) a mixed Mode I/Mode II fracture criteria.

The application of the damage tolerance methodology to the F-16 Production Fleet Management Program is described in reference 8. Test specimens used to generate the required Mode I and Mode II fracture toughness parameters are also described. This approach requires a significant analysis and test effort to evaluate "hot spots" within the structure and to generate the necessary fracture toughness data. The delamination growth assessment also requires a considerable 3-D finite element modelling and analysis effort. In addition, no reliable mixed mode fracture criteria has been reported. Hence, this approach is not considered sufficiently mature to warrant a recommendation for use in the certification of the X-Wing composite rotor blades.

## Safe Life (Reliability) Methodology

Statistically based certification methodologies provide a means for determining the strength, life, and reliability of composite structures. Such methods rely on the proper choice of population models and the generation of a sufficient behavioral database. Of the available models, the most commonly accepted for both static and fatigue testing is the two-parameter Weibull distribution. It is attractive for the following reasons.

1. The simple functional form is easily manipulated.

2. Censoring and pooling techniques are available.

3. Statistical significance tests have been verified.

The cumulative probability of survival function is given by

$$P_s(x) = \exp[(-x/\beta)^{\alpha}] \qquad (1)$$

For composite materials $\alpha$ and $\beta$ are typically determined using the maximum likelihood method. Also, the availability of pooling techniques is especially useful in composite structures test programs where tests conducted in different environments may be combined. Statistical significance tests are used in these cases to check data sets for similarity.

The following paragraphs present a review of the statistical certification method of reference 9. Related work is documented in reference 10. The development tests required to generate the behavioral database are outlined, followed by a discussion of the specific requirements for static strength and fatigue life testing. Special attention is given to the effect that matrix and fiber dominated failure modes have on test requirements.

A key to the successful application of a statistical certification methodology is the generation of a sufficiently complete database. The tests must range from the level of coupons and elements to full scale test articles in a "building block" approach. Additionally, the test program must examine the effects of the operating environment (e.g., temperature and moisture) on static and fatigue behavior. The coupon and subelement tests are used to establish material property variability. Although they typically focus on in-plane behavior, it is important to also include transverse properties. This is especially important in an application such as the X-Wing. The resulting data can be pooled as required and estimates of the Weibull parameters made.

Thus, the level and scatter of possible failure modes can be established. Transverse data is characterized by the highest scatter. Element and subcomponent tests can be used to identify structural failure modes. They may also be used to detect the presence of competing failure modes. Higher level tests, such as components, can be used to investigate structural response variability resulting from fabrication techniques. The resulting database should describe the failure mode, data scatter and response variability of a composite structure to a desired level of confidence. This data along with full scale test articles can be used to justify certification.

Out-of-plane failure modes can complicate generation of the database. Well-proven and reliable transverse test methods are few. The typically high data scatter makes higher numbers of tests desirable. Also, the increased environmental sensitivity in the thickness direction can cause failure mode changes, negating the ability to pool data and possibly resulting in competing failure modes. Thus, a design whose structural capability is limited by transverse strength can lead to increased testing requirements and certification difficulties.

The static strength of a composite structure is typically demonstrated by a test to Design Ultimate Load (DUL) which is 1.5 times the maximum operating load (Design Limit Load (DLL)). Figure 5 shows the reliability achieved for a single static ultimate test to 150% of DLL for values of the static strength shape parameter from 0 to 25. For fiber dominated failure, with $\alpha_S$ values near 20, such a test would demonstrate A-basis (99% probability, 95% confidence) reliability. However, for matrix dominated failure modes, with $\alpha_S$ ranging from 5 to 10, a test to 150% of DLL would not demonstrate A-basis. In fact, for values of $\alpha$ below 7, B-basis (90% probability, 95% confidence) reliability could not be demonstrated. Two options are available to increase the demonstrated reliability: (1) increase the number of test specimens or (2) increase the load level. The most effective choice is to increase the load level beyond 150% DLL, whereas increasing the number of test specimens yields little benefit and is expensive.

The two most applicable methods of statistical certification approaches for fatigue are the life factor (also known as scatter factor) and the load enhancement factor. The life factor approach relies on knowledge of the fatigue life shape parameter $\alpha_L$ from the development test program and a full scale test or tests. The factor gives the number of lives that must be demonstrated in test to yield a given level of reliability at the end of one life.

A plot of life factor $N_F$ versus fatigue life shape parameter $\alpha_L$ is given in figure 6 for a typical scenario. A single full scale test to demonstrate B-basis reliability at the end of one life is to be conducted. The curve shows that as the shape parameter approaches 1.0, the number of lives rapidly becomes excessive. Such is the case of in-plane fatigue failure ($\alpha_L = 1.25$). Although little data for transverse fatigue failures are available, it is reasonable to assume that the shape parameter will be the same or less. Hence, it is apparent that the life factor approach is not acceptable for certification of composites, especially where out-of-plane failure modes are dominant.

An alternate approach to life certification is the load enhancement factor, wherein the loads are increased during the fatigue test to demonstrate the desired level of reliability. Figure 7 illustrates the effect of the fatigue life shape parameter $\alpha_L$ and residual strength shape parameter $\alpha_R$ on the load enhancement factor required to demonstrate B-basis reliability for one life using a single full scale fatigue test to one lifetime. It is obvious that the required factor does not change significantly for fatigue life shape parameters in the range of 5 to 10. However, as $\alpha_L$ approaches 1.0, as it does for composites, the required load enhancement factor increases noticeably, especially for low values of the residual strength shape parameter. This curve illustrates well the potential problems that may arise from dominant out-of-plane failure modes. Such failure modes tend to have low values of $\alpha_L$ (near 1.0) and also low values of $\alpha_R$ (in the range from 5.0 to 10.0). These values would make the required load enhancement factors prohibitively large. It is evident that for failure modes which exhibit high static and fatigue scatter, the life factor and load enhancement factor approaches can result in impossible test requirements. A combined approach can be achieved through manipulation

of the functional expressions. The resulting method allows some latitude in balancing test duration and the load enhancement factor to demonstrate a desired level of reliability.

Figure 8 shows the curves of load enhancement factor versus life factor for the cases of fiber and matrix dominated failure. Typical values for the fatigue life and residual strength shape parameters are employed. The curves show the possible combinations of life factor (or test duration) and load enhancement factor to demonstrate B-basis reliability at the end of one lifetime using a single full scale fatigue-test article. The curve for fiber dominated failure modes exhibits quite reasonable values of life factor and load enhancement factor. For test durations ranging from 1 to 5 lifetimes, the load enhancement factor ranges from 1.18 down to 1.06. However, the test requirements for matrix dominated failure are more severe. Over the range of life factor from 1 to 5, the load enhancement factor ranges from 1.4 down to 1.19. An environmental knockdown factor would further complicate the test of a matrix dominated failure. Such a factor must be combined with the load enhancement factor to yield the required test load level. As is well known in composites, the adverse effects of environment on matrix properties is much more severe than on fiber dominated properties and the resulting factor may be significant.

The problems induced by matrix dominated failure can be further illustrated by assuming a limit exists on the load enhancement factor. Such limits may exist because of failure mode transitions at higher load levels. For instance, assuming a load enhancement factor of 1.2 is the maximum allowable, it is obvious that a successful one-lifetime test for a fiber dominated failure will demonstrate better than B-basis reliability. For matrix dominated failure, the same reliability would require a test duration of about 4.5 lives.

Two key aspects to the statistical certification methodology are the generation of an adequate database and the proper execution of a full scale demonstration test. The development test program must be conducted in a "building block" approach which produces reliable data on material shape parameters, environmental effects, failure modes and response variability. Perhaps the most important result is the ability to predict failure mode and know the scatter associated with it. Structures that exhibit transverse failures, which can result in competing modes and high scatter, may render the application of this failure methodology impractical. This result has been illustrated by the effect of shape parameters on both static and fatigue test requirements. The requirements clearly show that a well designed structure which exhibits fiber dominated failure modes will be more easily certified than one constrained by matrix dominated effects.

## Wear Out Methodology

The wear out methodology was developed in the early 1970s and is comprehensively summarized in reference 11 by Halpin, Jerina, and Johnson. This methodology was previously used in the certification of composite structural hardware such as (1) the A-7 outer wing, (2) the F-16 empennage, and (3) the B-1A horizontal tail.

In essence, the wear out approach recognizes the probability of progressive structural deterioration of a composite structure. The approach utilizes development test data on the static strength and the residual strength, after a specified use period, in conjunction with proof testing of all flight

hardware items to characterize this deterioration and protect the structure against premature failures. It is evident that the residual stiffness is an indicator of the extent of structural deterioration and can be an important performance parameter with regard to the frequencies of oscillation of flight surfaces.

The difficulties in implementation of the methodology include the determination of the critical load conditions to be applied for static and residual strength and stiffness testing and for the proof load specification. Similar difficulties would arise in the case of all candidate methodologies considered here and indeed emphasize the importance of a representative structural analysis. However, the advantage of the wear out approach for advanced composite hardware development projects, such as the X-Wing rotor blades, is the ability to assign "gates" for safe flight testing as the flight envelope is progressively expanded, i.e., for the stopped rotor flight phase, rotary wing flight phase, and finally conversion.

**Proof Test Philosophy–** The truncation in static and residual strength and life capacity resulting from proof testing is intended to develop confidence that the structure is unlikely to fail within a specified time under a specified usage. Most of the essential features of the wear out process are illustrated in figure 9. The structural deterioration can be represented by the following equation from reference 11:

$$F(t)^{2(r-1)} = F(0)^{2(r-1)} - (r-1)\, A_4\, (F_{max})^{2r}\, (t - t(0)) \qquad (2)$$

The key wear out parameter r is the slope of the da/dN curve or may be derived from the S-N fatigue curve for the failure mode in question. Based on this model, the proof load level required to protect the structure for the desired operating period t(op) can be deduced as follows:

Setting $F(0) = F(PL)$, and using $F(t) = F(xx)$, we obtain:

$$F(PL)^{2(r-1)} = F(xx)^{2(r-1)} + R\,t(op) \qquad (3)$$

where

$$R = (r-1)\, A_4\, (F_{max})^{2r} \qquad (4)$$

A minimum of two tests are required to determine the damage accumulation rate R: (1) a static test to failure and (2) a fatigue test followed by a residual strength test to failure. A residual stiffness test should also be performed for reasons noted earlier.

**Wear Out Model Database–** It is apparent from the above discussion that a reliable estimation of the damage accumulation rate R is the key to appropriate application of the wear out methodology. Since R essentially depends on the parameter r, data pertaining to the detailed configurations and failure modes of the hardware components in question must be obtained. We have observed that a distinct likelihood of matrix-controlled failure modes exists in the X-Wing rotor assembly and we therefore consider that failure modes encountered in bonded joint fatigue and in the delamination of advanced composite structures are most pertinent. Such data were derived from a number of

USAF-funded programs conducted in support of the development of the wear out model. These data are summarized in table I from J. C. Halpin (private communication).

As the data indicate, a close relationship between the wear out parameter r for bonded joints and composite laminates appears to exist. Furthermore, both of these generic features exist in the subject composite hardware. Unfortunately, no specific data for the fiber or the 5245C matrix system used for the rotor assembly could be found in the available literature.

## IMPLEMENTATION OF THE WEAR OUT METHODOLOGY

The application of the wear out methodology to developmental composite hardware projects such as the X-Wing rotor assembly seems to be feasible. The key tasks necessary to implement the methodology are depicted in figure 10 and can be summarized as follows:

1. Determine a best estimate of the usage spectrum, including duration t(op) or various multiples thereof to represent phases of the flight test program.

2. Based on an adequate structural analysis of the critical loading conditions, specify a static strength requirement and conduct a static strength test to failure for the condition deemed to be most critical.

3. Conduct a fatigue test based on the usage spectrum incorporating damage tolerance criteria backed by nondestructive inspection.

4. Conduct a residual strength and stiffness test to failure on the fatigued component drawing again on the static strength requirement defined earlier.

5. Estimate damage accumulation rate R through a series of tests of critical subelements (identified by the structural evaluation) and/or coupons. These tests should provide estimates of the wear out parameter r.

6. Conduct a proof test of each flight hardware component to a level deduced from items 1 through 5.

### Development of an Interlaminar Tension Test Coupon

To support the implementation of the wear out methodology for matrix dominated failures requires the generation of a database on the intrinsic wear out characteristics of the specific fiber-resin system in use. No existing interlaminar tension test specimen could provide the necessary flexibility for testing at the extreme environmental conditions the rotor blades would be exposed to. Consequently, Ames Research Center undertook development of an interlaminar tension test specimen to examine the validity of the wear out methodology for matrix tension failures.

The specimen configuration is shown in figure 11. It is a curved beam with all plys oriented around the circumferential direction (100% 0° layup). This type of ply schedule eliminates the free edge ILTS intensification that would occur for cross-ply laminates as load is applied or during cooldown from the cure temperature. The load P is applied at the free ends of each beam (flat section), which tends to pull them apart. This load acting over moment arm L generates maximum ILTS within the laminate at the apex (test section) of each specimen. Note that with curved beam geometries, the applied load can be introduced well away from the test section. The peak interlaminar tensile stress at the apex may be closely estimated by the following equation from reference 1:

$$\sigma_{zz_{max}} = \frac{3PL}{2bc\sqrt{R_i R_o}} \tag{5}$$

Finite element analysis has confirmed that equation (5) gives a very close estimation of the peak ILTS for simple geometries with small deflections. A more detailed analysis using classical elasticity (continuum) theory and multilayer (discrete) theory is given in reference 12.

A number of curved beam test specimens have been fabricated and tested to failure at Ames Research Center. These include the semicircular type described above and another variation having an elliptical shape. The initial results of static ultimate and fatigue to failure tests are reported in reference 13. Additional specimens are currently being fabricated to support residual strength testing and the determination of matrix wear out parameters.

## CONCLUSIONS

Three candidate fatigue-life methodologies were evaluated for use in certifying composite rotor blades on the RSRA/X-Wing Program. Delamination of the primary spar during ground testing had demonstrated the existence of critical interlaminar stress components. Analysis indicated that a number of regions were being exposed to significant out-of-plane load components.

The damage tolerance, safe life (reliability), and wear out methodologies were evaluated on the basis of their applicability to matrix dominated failure modes. The absence of reliable mixed mode fracture criteria made the use of the damage tolerance approach questionable. The statistically based safe life (reliability) method appeared practical for low scatter failures like those seen with isotropic metallics or fiber dominated modes, but the testing requirements for high scatter matrix dominated failure modes became impractical.

The chosen approach was the wear out methodology. This approach assumes that the structural degradation that occurs with use can be monitored by measuring parameters such as residual strength and stiffness. Proof testing prospective flight components to a predetermined static load can establish a safe envelope of operation for a specified number of cycles. This aspect allows existing rotor blades to be sequentially certified for flight conditions of increasing severity. For low load operating conditions (stopped rotor flight) the blades are proof tested to some fraction of design ultimate load.

Later on in the flight test program, the same blades can be proof tested to higher static load levels to certify the blades to more severe flight conditions such as conversion.

The wear out methodology requires an extensive matrix of testing to determine the failure characteristics of the components. Analysis is used to determine the failure prone "hot spots" in the structure and static and fatigue tests are conducted on full scale hardware to establish the locations and types of failure modes and the rates of damage propagation. Extensive subelement and coupon tests are conducted to make estimates of the damage accumulation rate for the particular failure mode and portion of the structure represented by the subelement. From these data, proof load levels can be established for flight components to assure their integrity for the duration of the operating period.

Testing is underway at Ames Research Center utilizing a new interlaminar tension test coupon to assess the validity of the wear out methodology for matrix tension failure modes.

## REFERENCES


1. Kedward, K. T.; Wilson, R. S.; and McLean, S. K.: The Flexure of Simply Curved Composite Shapes. Composites. vol. 20, no. 6, November 1989, pp. 527-536. ASTM STP 787, American Society for Testing and Materials, 1982, pp. 5-18.

2. Bohon, H. L.; Chapman, A. J. III; and Leybold, H. A.: Ground Test Experience with Large Composite Structures for Commercial Transports. NASA TM-84627, March 1983.

3. Hoa, S. V.; Daoust, J.; Du, B. L.; and Vu-Khanh, T.: Interlaminar Stresses in Tapered Laminates. Polymer Composites, vol. 9, no. 5, October 1988, pp. 337-344.

4. Paul, P. C.; Saff, C. R.; Sanger, K. B.; Mahler, M. A.; Kan, H. P.; and Kautz, E. F.: Out of Plane Analysis for Composite Structures. Eighth DOD/NASA/FAA Conference on Fibrous Composites in Structural Design, Norfolk, Virginia, 1989.

5. Lane, J. W.; and Sumich, M.: The RSRA/X-Wing Experiment: A Status Report. 1987 SAE International Powered Lift Conference and Exposition, Santa Clara, Calif, Dec. 1987.

6. Rogers, C.: Interim Report on Design Criteria for Damage Tolerant Helicopter Primary Structure of Composite Materials. Contract DAAK51-84-C-002.

7. Rybicki, E. F.; and Kanninen, M. F.: A Finite Element Calculation of Stress Intensity Factors by a Modified Crack Closure Integral. Engineering Mechanics, 1977, vol. 9, pp. 931-938.

8. Wilkins, D. J.: A Preliminary Damage Tolerance Methodology for Composite Structures, Workshop on Failure Analysis and Mechanisms of Failure of Fibrous Composites. NASA Langley Research Center, 1982.

9. Whitehead, R. S.; Kan, H. P.; Cordero, R.; and Saether, E. S.: Certification Testing Methodology for Composite Structures. Northrop Corporation, Aircraft Division, NADC-87042-60, Oct. 1986.

10. Sanger, K. B.: Certification Testing Methodology for Composite Structures. McDonnell Aircraft Corporation. Report NADC-86132-60, January 1986.

11. Halpin, J. C.; Jerina, K. L.; and Johnson, T. A.: Characterization of Composites for the Purpose of Reliability Evaluation, Analysis of Test Methods for High Modulus Fibers and Composites. ASTM STP 521, American Society for Testing and Materials, 1973, pp. 5-64.

12. Ko, W. L.; and Jackson, R. H.: Multilayer Theory for Delamination Analysis of a Composite Curved Bar Subjected to End Forces and End Moments. NASA TM-4139, September 1989.

13. Hiel, C. C.; Sumich, M.; and Chappell, D. P.: A Curved Beam Test Specimen for Determining the Interlaminar Tensile Strength of a Laminated Composite. Eighth DOD/NASA/FAA Conference on Fibrous Composites in Structural Design, Norfolk, Virginia, 1989.

**Table 1. Relevant data providing indicators of damage accumulation rate R by use of wear out parameter r**

Fatigue of bonded joints

| Configuration | Source | Estimated r |
|---|---|---|
| Scarf joint | General Dynamics | 2.5 |
| Double-lap joint | General Dynamics | 2.2 |
| Double-lap joint | Northrop | 2.5 |
| Double-lap joint | NASA/LARC | 3.3 |

Fatigue of composite laminates

| Construction | Source | Estimated r |
|---|---|---|
| $[\pm 45]_s$ Boron/epoxy | General Dynamics | 5.1 |
| [0] F–16 Graphite/epoxy | General Dynamics | 4.0 |
| $[0,90]_s$ F–16 Graphite/epoxy | General Dynamics | 3.4 |

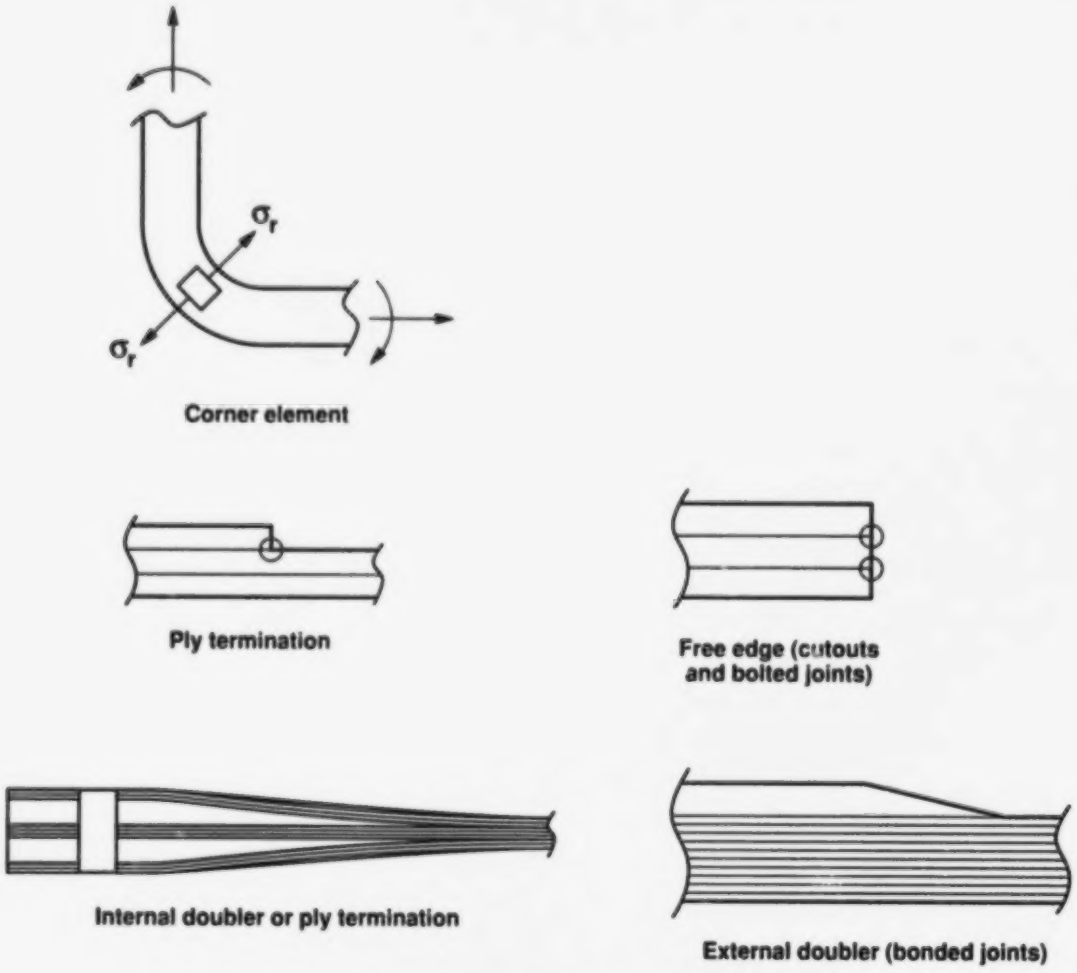


Figure 1. Sources of delamination stresses.

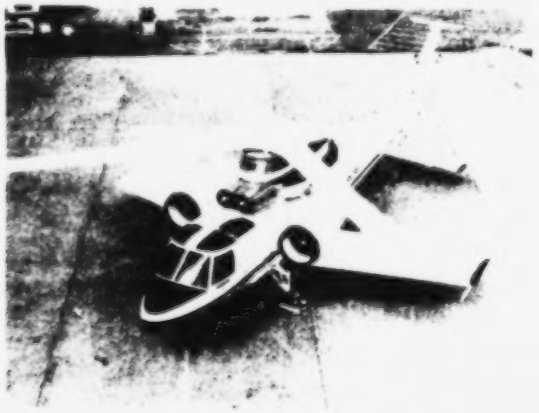

Figure 2. RSRA/X-Wing flight vehicle.



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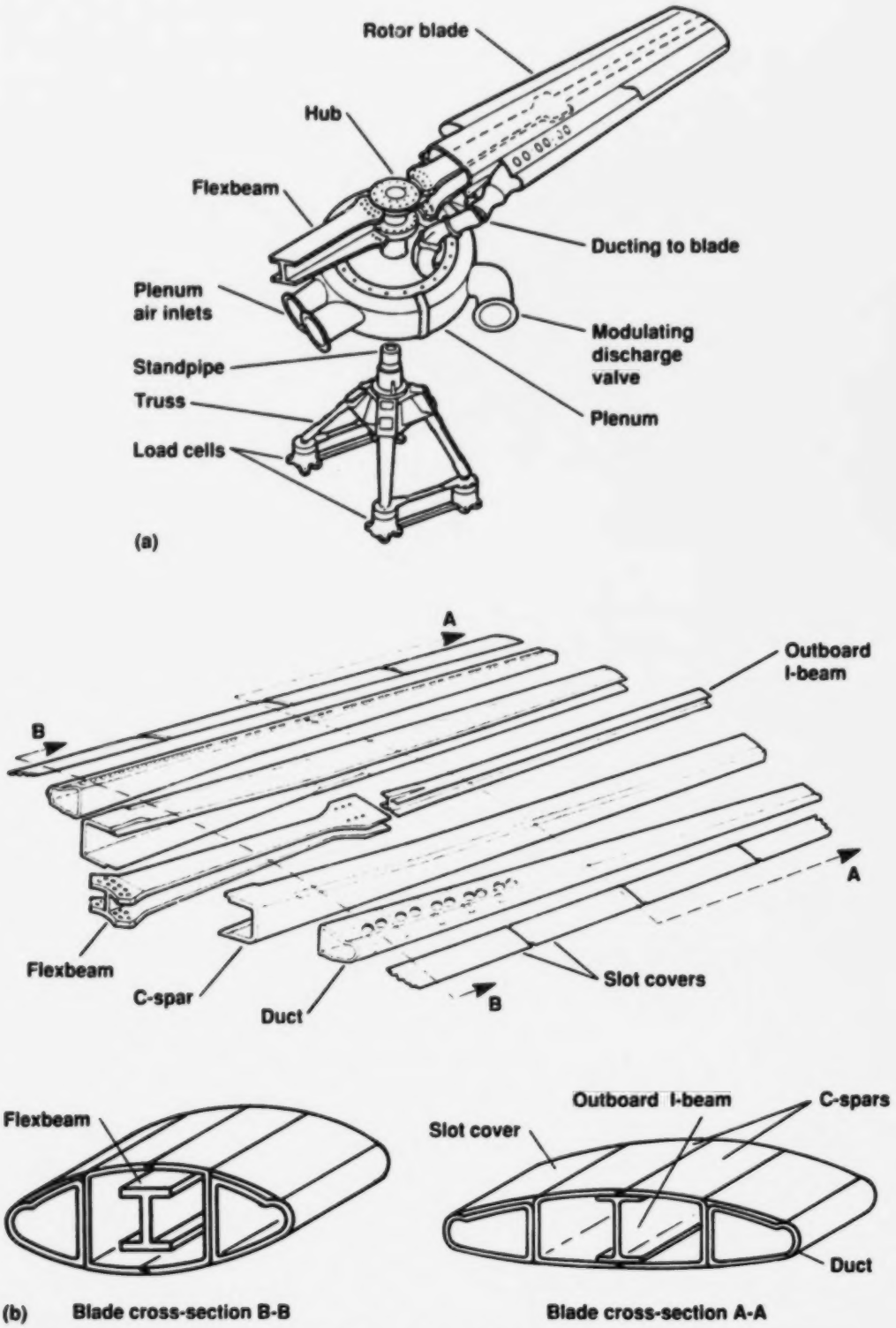


Figure 3. X-Wing rotor system. (a) Rotor system schematic; (b) major rotor blade components and cross-sections.

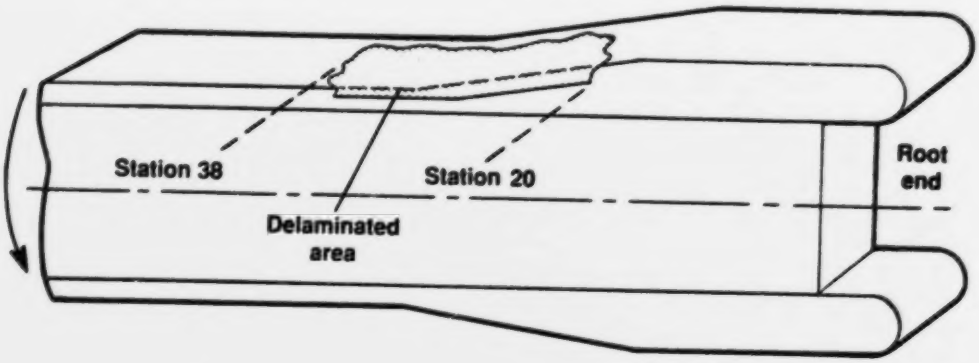


Figure 4. Delaminated region of failed flexbeam.

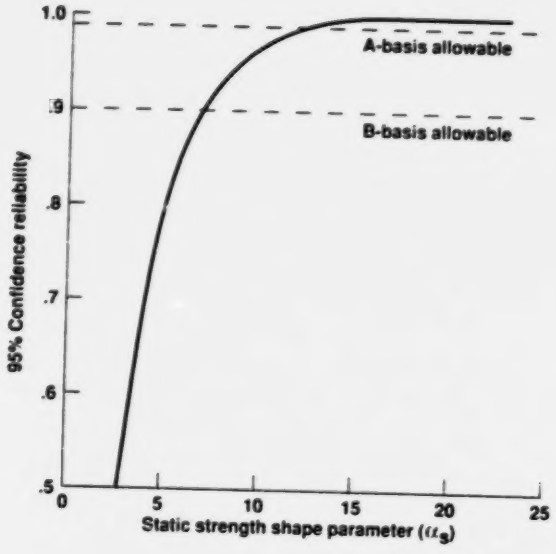


Figure 5. The 95% confidence reliability versus the static strength shape parameter ($\alpha_s$) for a single full scale static test to 150% of design limit load.

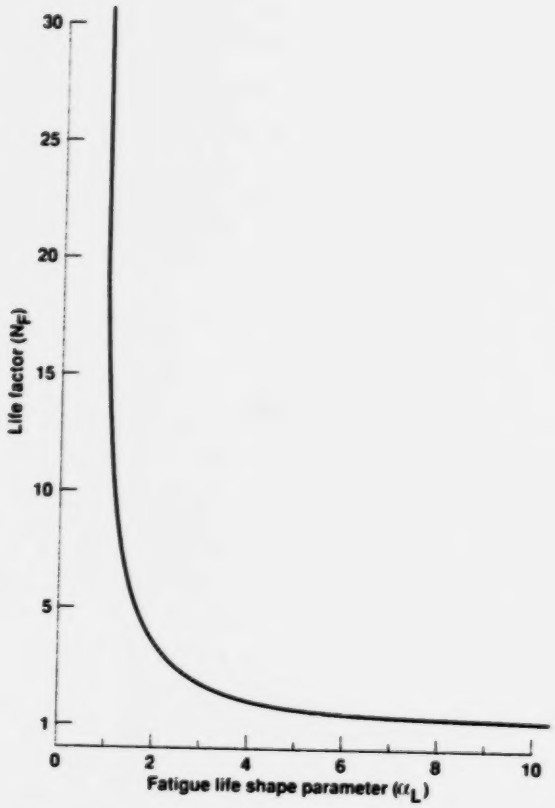


Figure 6. Life factor ($N_F$) required to demonstrate B-basis reliability at the end of one life versus fatigue life shape parameter ($\alpha_L$) using a single full scale fatigue test article.

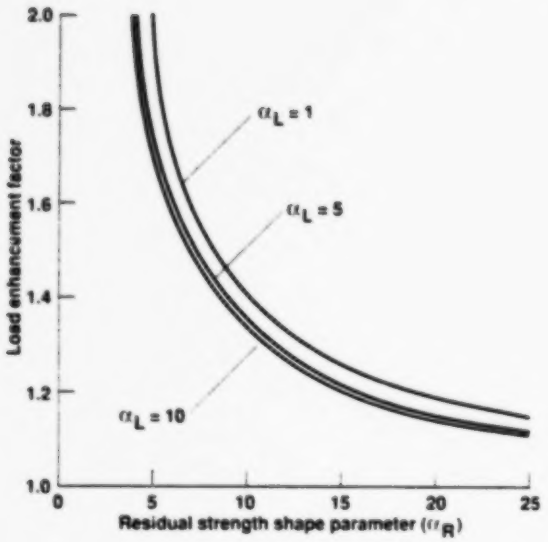


Figure 7. Load enhancement factor (required to demonstrate B-basis reliability at the end of one life) versus the residual strength shape parameter ($\alpha_R$) using a single fatigue test to one lifetime.

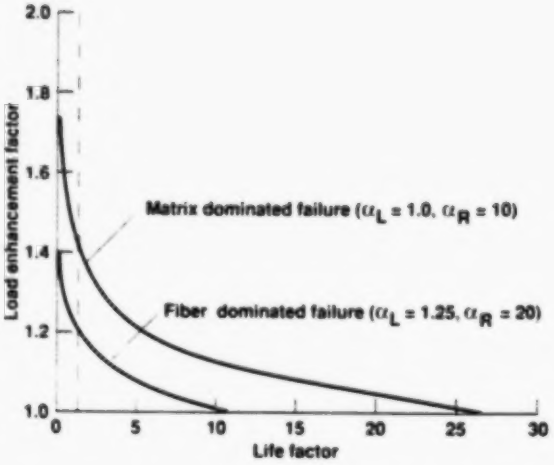


Figure 8. Possible combinations of load enhancement factor and life factor ($N_F$) (necessary to demonstrate B-basis reliability at the end of one lifetime) using a single full scale test article.

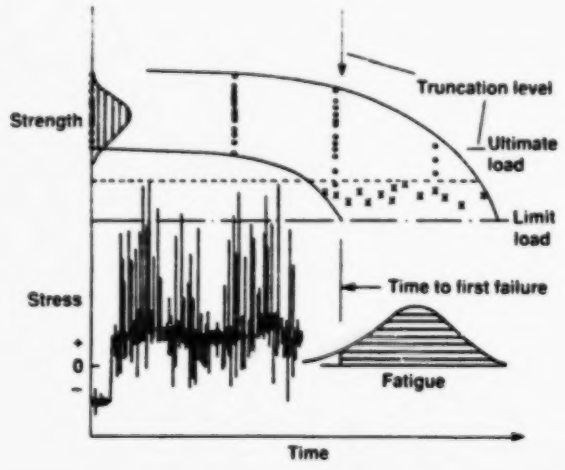


Figure 9. Wear out model showing the relation of static failure, load history, and fatigue failure (ref. 11).

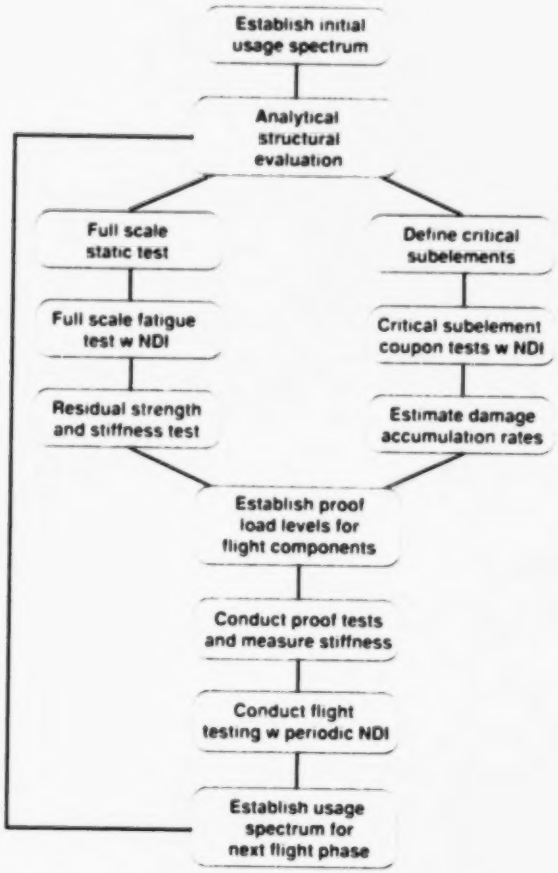
Establish initial usage spectrum
Analytical structural evaluation
Full scale static test
Define critical subelements
Full scale fatigue test w NDI
Critical subelement coupon tests w NDI
Residual strength and stiffness test
Estimate damage accumulation rates
Establish proof load levels for flight components
Conduct proof tests and measure stiffness
Conduct flight testing w periodic NDI
Establish usage spectrum for next flight phase

Figure 10. Application of the wear out methodology.

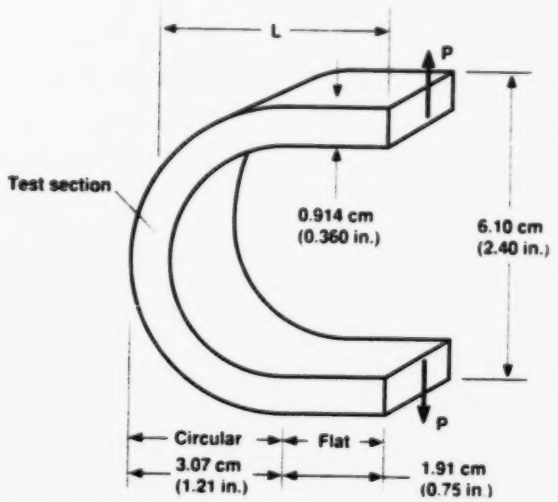


Figure 11. Curved beam interlaminar tension test coupon.

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15 Supplementary Notes

Point of Contact: Mark Sumich, Ames Research Center, MS 237-7, Moffett Field, CA 94035-1000
(415) 604-6193 or FTS 464-6193

16 Abstract This report documents the efforts of the RSRA/X-Wing Project Office to identify and implement a fatigue-life methodology applicable to demonstrated delamination failures for use in certifying composite rotor blades. The RSRA/X-Wing vehicle was a proof-of-concept stopped rotor aircraft configuration which used rotor blades primarily constructed of laminated carbon fiber. Delamination of the main spar during ground testing demonstrated that significant interlaminar stresses were being produced. Analysis confirmed the presence of out-of-plane load components. A review of the available failure methodologies was undertaken to determine the approach most applicable to certifying primary composite structures which can fail via delamination. The final selection of the "wear out" (residual strength) methodology and the requirements for its implementation are discussed. Composite structural designs incorporating out-of-plane load components are not endorsed. However, a means of certifying development hardware is necessary when an identification of matrix dominated failure modes occurs late in the development cycle. Efforts are continuing at NASA Ames Research Center to assess the validity of the "wear out" based failure methodology for interlaminar tension failures. A different type of test coupon has been developed and testing is underway to establish a database on the "wear out" characteristics of a composite structure subjected to out-of-plane loads.

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